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QUALIFICATION AND TESTING OF AN ELECTRICALLY PROPELLED SPACECRAFT - SERT II

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SUMMARY

Development of a spacecraft using ion thrusters designed to operate for 6 months entailed unusual qualification problems. Testing with ion thrusters required vacuum operation. The spacecraft was in the same chamber with a plasma created by the thruster. High voltages required for ion thrusters may cause electrical breakdowns or cause the spacecraft and Aerospace Ground Equipment (AGE) to receive high power transients. Also, components that were qualified for a 6-month mission were scarce. This report documents the resolution of these problems. Since the program scheduled only a single flight, an extensive testing and development program was conducted to launch and successfully operate the spacecraft with one attempt.

INTRODUCTION

Ion thrusters are recognized as a propulsion technique of the future which provides low levels of thrust for long periods of time using little propellant. The SERT II system operates at 0.0063-pound (0.028-N) thrust and a specific impulse of 4200 seconds. Such a system can be developed for thrusting on interplanetary missions and in attitude control and station keeping applications.

The ability of ion thrusters to function has been repeatedly demonstrated in ground tests at NASA Lewis Research Center and elsewhere. A space demonstration of electric propulsion was made in the SERT I mission wherein a thruster was operated for 30 minutes in space, demonstrating that an ion beam could be neutralized in space and that thrust could be produced. Reference 1 describes the ion thruster performance during that flight test. Reference 2 describes the SERT I spacecraft and its operation.

The major requirement for the SERT II spacecraft was to demonstrate the endurance capability of ion thrusters in the space environment. Since only a single flight was allo-

cated to the program, a conservative approach to qualification and testing was required to ensure a high probability for success on the initial mission attempt.

The qualification program of the SERT II spacecraft was typical of other spacecraft development programs. It involved component qualification and system qualification to expected vibration and shock levels as well as to expected thermal and vacuum environments. In addition, however, more specialized problems associated with the electric propulsion system integration were faced. Primary areas of concern were (1) operating the spacecraft in a vacuum chamber containing the ion thruster plasma efflux, (2) operating the power conditioner (which produces +3000V and -2000V dc) on the spacecraft and controlling breakdowns, and (3) demonstrating endurance capability of the spacecraft design to survive a 6-month space mission.

This report describes the vibration and vacuum test facilities used to test and qualify the integrated spacecraft systems. It also describes the test programs carried out on the four spacecraft models: namely, the experimental mass dummy, the experimental operational model, the prototype, and the flight model.

DESCRIPTION OF TEST FACILITIES AND PROCEDURES

General

Preliminary testing on each operating spacecraft model was done in air under standard laboratory conditions. Special facilities were required to perform vibration and shock testing and vacuum testing on the integrated systems. The experimental operational model was tested under thermal vacuum conditions but not vibrated; the experimental mass dummy model was vibration and shock tested but not vacuum tested. The prototype and the flight model were each subjected to both environments.

Vibration Testing

Vibration and shock testing were carried out on a vibration generator which had a capability of 28 000 pounds (1.25×10^5 N) force. For thrust axis vibration, the spacecraft model was mounted to the shake fixture, and the fixture was suspended by shock (bungee) cord so that the spacecraft and fixture weight was supported by the cord instead of resting on the vibration generator. The shaker force capability then was sufficient to produce the qualification levels. Figure 1 shows the prototype spacecraft mounted on the vibration generator being prepared for the thrust axis vibration. Notice the magnesium casting adaptor which adapted the spacecraft mounting face to the shaker. This figure also shows provisions for mounting the spacecraft for lateral testing. The spacecraft for lateral test was mounted on a magnesium casting which rested on four vibration, or slip,

tables. These were hydrostatic bearings using hydraulic fluid, under pressure, to provide minimal resistance to lateral motion. The vibration generator was pitched over 90° on its trunnions to drive the system in the lateral test modes.

The vibration generator was also used to shock test the spacecraft models with half-sine pulses of 8 milliseconds duration in each direction on each axis.

Vacuum Testing

Vacuum testing of the spacecraft was carried out in a vacuum chamber which is 15 feet (4.6 m) in diameter and 60 feet (18.3 m) long. It is capable of maintaining a pressure of 1×10^{-7} torr. The tank is the 15-foot model described in detail in reference 3. Testing the spacecraft and ion engines in a vacuum tank is more severe in many ways than operating the system in space.

Outgassing. - The vacuum attained in the vacuum chamber is not as "hard" as that prevailing in space; therefore, high-voltage arc-over problems are aggravated. Also, outgassing of components creates local pockets of relatively high pressure where arcs can occur. These problems were solved with proper testing procedures. A procedure was evolved in which the test system was vacuum baked for 12 hours before test each time the system had been exposed to air. During this outgassing period, the power conditioner (where high-voltage problems were most severe) was maintained at 165° F (74° C) to facilitate outgassing. After this vacuum soak, the system was cooled to the operating temperature range and testing was performed.

Floated operation. - Another provision to simulate space operation in the vacuum tank was to electrically isolate the spacecraft with the ion thruster from all surrounding structures. If this was not effected properly, the neutralizer system of the thruster would not receive an adequate test. The ion thruster produces thrust by expelling heavy mercury ions which are positively charged. An equal number of negative charges in the form of electrons must also be expelled to keep the potential of the spacecraft from going negative and attracting the mercury ions, thus stopping thrust. Supplying the required electron flow is the function of the neutralizer system. If the spacecraft and ion thruster are grounded, the supply of electrons would come partially from earth ground rather than the neutralizer; consequently, the neutralizer would not be adequately tested. For this reason, the spacecraft was usually operated in the floated mode.

This constraint resulted in a mechanical installation which allowed the spacecraft to be electrically isolated from the vacuum chamber. Figure 2 shows the spacecraft installed in the chamber. It was supported by trolleys on an overhead beam assembly. The spacecraft was suspended from steel cables which had insulators installed. The spacecraft could then be electrically isolated (floated) from the tank structure to more

closely simulate space conditions. Figure 3 shows the experimental spacecraft on the insulated suspension at the entrance to the tank.

Electrical installation. - To complement the floated mechanical installation, steps were taken to ensure that no spacecraft grounds were introduced through the power supply system of the electrical installation. Safety of personnel was also considered in configuring the electrical equipment. Due to the possibility of high voltages being generated on the spacecraft and applied to the AGE, the dc power supplies for operating the spacecraft were enclosed in, and isolated from, a wire-mesh enclosure. This prevented personal injury if the supplies should float to high voltages. These supplies were also isolated from the ac power line by isolation transformers to further limit potential damage due to high voltage and to isolate the system from ground. Control of the supplies and other equipment was accomplished by using insulated control knobs extending to the outside of the enclosure or by using high-voltage relay control circuits. The electrical installation of the spacecraft is shown schematically in figure 4. The three power supplies shown represent the three sections of the SERT solar array. The thruster and power conditioner were operated from the nominally 56 volts of the two series-connected supplies. The remaining spacecraft systems (called housekeeping systems) were operated from the third supply. The series resistors shown in the output of the three supplies provided an output characteristic similar to the static solar array voltage-current characteristic which powers the spacecraft in flight. Figures 5 and 6 show these characteristics. This feature could be bypassed, when desired, by shorting the resistors using relay contacts K_3 and K_4.

One of the electrical cables leading from the spacecraft to the outside of the tank was attached to the spacecraft single-point ground. This lead was "floated" or "grounded" to the tank structure as specific tests demanded. This is the same cable that was connected to the Agena vehicle ground point for flight. Figure 4 shows the capability of floating or grounding the spacecraft using relay K_1 or of floating or grounding the ion beam target using relay K_2. Reference 4 contains a discussion of floating and grounding ion engine systems. In this program the spacecraft was usually floated and the target grounded during test operations.

Contamination shield. - In space, the ion beam and the electrons from the neutralizer recombine and diffuse into the infinite space environment. Whereas, in the finite volume of the vacuum chamber, the efflux from the ion thruster impinges on the grounded target and produces sputtered material which ultimately plates surfaces in the tank. In order to protect equipment and the end of the tank downstream of the thruster, a 6-foot- (1.8-m-) square stainless-steel sheet was installed as a target. This target was usually grounded during operations so that the ion beam terminated in an all-grounded structure of target and chamber walls.

Also, recombination of the ions and electrons in the tank produced mercury atoms which condensed on material in the tank. It was also theorized that some free ions and

electrons existed in the tank during operation, which could cause arc-overs within the spacecraft if the plasma migrated to critical areas. In an attempt to lessen the effect of these ground test limitations, a contamination shield was developed. It had the primary purpose of blocking sputtered target material from plating onto the spacecraft. It also prevented ion engine plasma efflux from engulfing the spacecraft. The center portion of the shield was fabricated from solid stainless-steel sheet to shadow shield the spacecraft from sputtered material. Holes were cut for the ion beam to pass through. Number 40 grid, stainless-steel mesh was used for the outer portion of the shield as a plasma barrier. This shield assembly was usually grounded to the tank structure. Figure 7 shows the experimental spacecraft in the vacuum chamber with the contamination shield in place.

Radiofrequency link testing. - Commands to the spacecraft and data from the spacecraft during testing were accomplished by radiofrequency (RF) link. This mode of operation reduced the number of required "hardlines" to the spacecraft and thereby simplified the effecting of floated operation. It also gave early experience with the flight-type command and telemetry systems. The total system schematic for operating the spacecraft in the vacuum facility is shown in figure 8.

Spacecraft control center. - In the experimental test program phase, control of the spacecraft operations was based in an 8 foot by 20 foot (2.4 m by 6.1 m) communications van. This van housed the command encoder and transmitter. It also contained the data readout systems consisting of telemetry receivers, tape recorder, discriminators, decommutator, and signal conditioning. Data output was on a high-speed printer and pen recorders. In order to monitor continuous tests, a meter panel with 32 set-point-type meters was used to monitor critical system parameters continuously and to trigger an alarm when preset limits were exceeded.

During the prototype program, control was transferred to the permanent control center - the one that ultimately would monitor the spacecraft in orbit. A computer to limit check the data was acquired and integrated into the system. Its data output was on three teletypes. Teletype data output was automatically flagged whenever preset limits were exceeded. A display was used to supplement teletypes. It contained an illuminated display of 25 key parameters with red or green lights continuously indicating status.

As launch approached, data center rehearsals were conducted. Some rehearsals were conducted using a spacecraft data simulator, and some used the spacecraft in the vacuum tank while orbital operations and pass durations were simulated. In this way all subsystem engineers were familiar with flight-type operations and data format before launch. The chief difference between rehearsal and flight operations was that, after launch, commands are sent by the remote tracking stations of the NASA Satellite Tracking and Data Acquisition Network (STADAN) rather than from the spacecraft control center. Figure 9 shows the data center configuration in the late stages of the test program.

A teletype system to connect the control center to the STADAN network had been added by the time of launch.

Thermal environment control. - The thermal control equipment shown in figure 10 was used to control spacecraft temperature levels during tests. The vacuum chamber was lined with a black-painted, copper, cold wall which was filled with liquid nitrogen and held at -320° F (-195° C) to simulate the coldness of space.

In order to apply heat to the spacecraft sun side, a solar heat input plate was made which extended 150° around the sun side and along the length of the spacecraft. This curved plate was mounted beneath the spacecraft and was heated from below by quartz lamps. It was physically isolated from the spacecraft. The plate temperatures were monitored by automatic temperature controllers which cycled the quartz lamps. Heat flow into the spacecraft sun side was effected by radiation from the upper surface of the solar heat input plate to the spacecraft sun side.

The thermal interface with the Agena vehicle was simulated by mounting a flight-type shroud transition ring to the aft face of the spacecraft and mounting a 20-inch- (50.8-cm-) long, black-painted, cylindrical body to simulate the vehicle. This cylindrical section had four sets of resistance heaters mounted at 90° locations around the circumference. The heaters were electrically isolated from the surface so the spacecraft could be floated while the heaters remained operational. Manually controlling the voltages to each of the four sets of heaters permitted simulation of the heat input to the spacecraft from the Agena vehicle. A foil radiation shield was mounted aft of the spacecraft to limit the heat loss through the aft end of the simulator.

The thrusters mounted on the spacecraft were located forward of the solar heat input baffle and, therefore, received no heat input. The thrusters were exposed to the cold walls so provision was required to prevent the mercury propellant from freezing when the thrusters were not operating in the tank. Two quartz lamps were mounted near each thruster to radiate heat into each unit. These thruster heat lamps were manually controlled by adjusting input voltage to the lamps.

Thermal control of the power conditioners was maintained by four 500-watt heat lamps directed at the power conditioner bay on the space-facing side of the spacecraft. With the lamps off, this side of the spacecraft viewed the tank cold wall. The lamps were turned on to raise temperatures as needed. These lamps were manually controlled in pairs so that precise temperature levels could be controlled on the power conditioners.

DESCRIPTION OF SPACECRAFT TEST PROGRAMS

Experimental Program

The experimental test program was carried out using two spacecraft models. One was a mechanical, mass-dummy model used for vibration and shock testing. The other model was the experimental operational model used for electrical systems testing.

Experimental mass dummy model. - An experimental spacecraft structure was fabricated and loaded with component mass dummies. It was used to demonstrate the ability of the structure to meet the vibration and shock environment. This same structure was also used as a vibration fixture to qualify components for the prototype and flight units. This ensured more realistic testing and eliminated the need for separate fixtures for the components. The components under test were mounted in a flight-type configuration while the vibration input to the base of the spacecraft was controlled to levels proportional to expected vehicle output levels. This mass-dummy spacecraft survived an estimated 7 hours of vibration and shock testing with minor periodic repairs. This spacecraft was not capable of operating electrically but was only a mechanical model.

Experimental operational model. - An experimental spacecraft was prepared early in the program. Its prime purpose was to allow integrated electrical tests to be conducted to demonstrate the compatibility between power and communications systems and the ion thruster system. In assembling the experimental spacecraft, assembly and harnessing techniques were developed which were used in prototype and flight systems assembly. The experimental test program also served to refine Aerospace Ground Equipment (AGE) and facilities required later to test prototype and flight spacecraft.

The experimental spacecraft was fabricated as close electrically to the flight design as could be achieved in the limited time available. It was assembled using commercially available electronic and electrical parts and without formal Reliability and Quality Assurance surveillance. The experimental spacecraft did achieve the required goals. This spacecraft unit was an electrical model, packaged similar to the flight design; however, it was not subjected to vibration and shock testing.

Experimental electrical testing. - The experimental spacecraft was checked out with integrated testing in an ambient air environment followed by a vacuum test to demonstrate that the housekeeping systems operated properly in a vacuum. Later, an ion thruster was added to the spacecraft. Thruster compatibility with the power and communications systems was demonstrated. During the early experimental testing, a flight-type packaged power conditioner was not available, so a rack-mounted breadboard unit located outside the vacuum chamber was used. The several voltages to operate the ion engine, including the high voltages, were carried into the vacuum chamber to the ion

engine via high-voltage feedthroughs in the tank wall. This early testing showed that the presence of the plasma in the tank did not preclude operation of this spacecraft.

Later, after completion of the first thermal test described below, a flight-sized power conditioner was available for integration into the spacecraft. The first test on the complete system was conducted with the spacecraft grounded. The vacuum test lasted for $2\frac{1}{2}$ hours of erratic power conditioner - thruster operation and it ended with a power conditioner diode failure. Severe arcing took place during the test and the telemetry system subcommutators were sampling data in improper, indeterminate sequence much of the time, so much data was lost. Subsequent tests with thruster and power conditioner operating were also erratic and short. Some solid-state devices in the subcommutators were damaged by high-current spikes being introduced on the data input lines during arcing. This problem on experimental and prototype spacecrafts led to the development of a filter box to protect the subcommutators. The failure mode was not repeated after addition of the filter box.

Thermal testing. - After thruster integration, but before power conditioner integration, a test was performed to verify the thermal design. The test objectives were (1) to determine the thruster heat inputs to the spacecraft, (2) to evaluate the analytical techniques used to determine the thermal design, and (3) to calibrate and operate the thermal control equipment of the facility. In order to perform the test, 150 thermocouples were installed on the spacecraft to measure temperature distributions. The presence of these thermocouples required the test to be run with the spacecraft grounded.

In general, the test showed that the thruster, mounted on its gimbal system, rapidly reached equilibrium temperature independently of the spacecraft. This indicated that the conduction through the gimbal system was negligible.

The analytical computer techniques were shown to be useful; however, the analytical model required more detailed treatment of components. Accuracy demonstrated was $\pm 15^{\circ}$ F ($\pm 8.3^{\circ}$ C), so work was also needed to improve accuracy of predictions. This test added urgency to an expanded computer model being developed at that time. The thermal control equipment did perform satisfactorily for this test.

After the power conditioner had been integrated into the spacecraft, a second thermal test was run. The objectives of this test were (1) to determine the effect of the power conditioner on temperature distribution, (2) to recheck the thruster heat inputs to the spacecraft, and (3) to recalibrate the facility and to recheck the analytical model. The spacecraft thermal control pattern had been changed significantly after the previous test and an updated computer model was used for predictions. The Agena vehicle thermal simulator had been added for this test also.

The objectives of this second thermal test were generally met. There were still some discrepancies between the thermal test data and the predictions, but they were eventually reconciled by compensating for test facility effects not applicable to the space flight case.

Endurance testing. - The experimental spacecraft continued to be operated to gain operational experience and to exercise all conceivable operating modes of the unit. It was desirable to accumulate as much test time as possible on the system components. Ultimately the experimental unit logged 500 hours of operation, of which 375 hours were with an operating ion thruster. The last test of the experimental spacecraft was an endurance test which accumulated 192 hours of uninterrupted thruster - power conditioner operation. This spacecraft was phased out when the prototype unit became available for test.

Prototype Program

The key principle in developing the prototype spacecraft was that it was maintained mechanically and electrically as identical to the flight spacecraft as possible. Its components were prepared for spacecraft assembly in the same manner as flight components. Component designs were selected and approved for inclusion in the prototype and flight units upon satisfaction of a qualification criteria specified for each component.

Component qualification criteria. - Whenever possible, equipment was specified which had successful flight histories on previous missions. These items were considered qualified by similarity to qualified equipment. On some newly designed equipment, qualification was established by test. In these cases, a unit was prepared and subjected to environmental testing to stress levels in excess of those expected in flight. This was to establish confidence in the component's design.

Some components were not completely qualified by similarity to qualified equipment or by testing. In these cases it was impractical to perform a complete qualification test program, so only a partial program was carried out. For instance, several items received only thermal tests, rather than thermal vacuum tests, by vendors with facilities limitations. In these cases, qualification was completed by being mounted on the prototype spacecraft during its qualification. This last approach to qualification was not as conservative as the first two methods. Some judgment was required to allow specific items into this category. Consideration was given to component flight histories and possible failure modes as well as to the impact of premature failure on the prototype spacecraft qualification effort. In the final analysis, all components mounted on the prototype spacecraft throughout its qualification test program were considered to be design qualified by virtue of successful conclusion of that program.

Component flight acceptance testing. - Each component to be used for the prototype or flight spacecraft or backup was subjected to flight level environmental testing to demonstrate that the specific component was ready for flight.

Flight acceptance tests for each component were specified in the Test Requirements Document for that article, which included the specification of testing required to ready that component for spacecraft assembly. Typically, the steps included were (1) electrical performance test, (2) vibration test in three axes, (3) shock tests in three axes, (4) electrical performance test, (5) thermal vacuum test, (6) special environmental tests, and (7) final electrical performance test. To sum up then, the prototype components, the flight components, and the flight backup components, as applicable, were all subjected to the same test program prior to installation on a specific spacecraft and could be assigned interchangeably.

Vibration and shock testing. - After assembly and electrical integration testing, the prototype spacecraft was mounted on the vibration generator described previously and subjected to vibration testing. Testing was done on the prototype spacecraft complete with squibs and a titanium gas sphere pressurized with a flight load of 3600 psi (2.48×10^{7} N/m^{2}) of Freon-14. The power to the spacecraft was turned off during vibration, but the control moment gyro rotors, which had been previously run up, were coasting, simulating the launch condition. Testing was done first on the thrust axis in the following sequence: sinusoidal test, random test, and shock test. An electrical checkout was performed and the system was reconfigured for the lateral tests in each of two axes. Tables I to IV give the vibration and shock levels imposed at the mounting face of the

TABLE I. - SINUSOIDAL VIBRATION LEVELS FOR THRUST AXIS

Frequency, Hz	Prototype model		Flight model	
	Acceleration, g's (zero to peak)	Sweep rate, octaves/min	Acceleration, g's (zero to peak)	Sweep rate, octaves/min
5 to 10	(a)	2	(a)	4
10 to 13	2.3		1.5	4
13 to 22	4.6		4.1	2
22 to 400	2.3		1.5	4
400 to 500	2.3 to 4.5		1.5 to 3.0	4
500 to 2000	4.5		3.0	4
Sweep time, min	4.3		2.15	

[a]Acceleration, 0.25 in. (0.64 cm) double amplitude displacement.

TABLE II. - SINUSOIDAL VIBRATION LEVELS FOR EACH OF TWO LATERAL AXES

Frequency, Hz	Prototype model		Flight model	
	Acceleration, g's (zero to peak)	Sweep rate, octaves/min	Acceleration, g's (zero to peak)	Sweep rate, octaves/min
5 to 10	(a)	2	(a)	4
10 to 250	1.5	↓	1.0	↓
250 to 400	3.0	↓	2.0	↓
400 to 500	3.0 to 4.5	↓	2.0 to 3.0	↓
500 to 2000	4.5	↓	3.0	↓
Sweep time, min	4.3		2.15	

[a]Acceleration, 0.25 in. (0.64 cm) double amplitude displacement.

TABLE III. - RANDOM VIBRATION LEVEL APPLIED TO THRUST AXIS AND EACH OF TWO LATERAL AXIS

Frequency, Hz	Prototype model		Flight model	
	Acceleration, g's rms	Spectral density, g^2/Hz	Acceleration, g's rms	Spectral density, g^2/Hz
20 to 400	3.32	0.03	2.21	0.015
400 to 2000	9.64	.06	6.43	.03
Total level	10.2	----	6.8	----
Duration, min/axis	4.5		2.25	

TABLE IV. - SHOCK PULSE LEVELS

[Half-sine pulses for 8 msec each; shock applied three times in each direction on each axis.]

Axis	Prototype model	Flight model
	Shock, g's	
Thrust	14	10
Lateral (each of two)	14	9

spacecraft. The specification levels were evolved from available Thorad-Agena flight data. Testing was most severe in the 13- to 22-hertz band due to known thrust oscillations of the Thorad main engine near burnout (commonly referred to as pogo effect). No electrical equipment failures occurred as a result of vibration and shock testing; however, mechanical failure of a thruster gimbal mechanism consisting of pyrotechnic pin pullers, flex pivots, and gimbal actuators did occur, resulting in unconstrained motions of a thruster on the spacecraft. The thruster gimbal assembly was subsequently redesigned to improve the pin puller strength. Also, the flex pivots were replaced by

conventional bushings. This redesigned system was qualified as a component using the experimental mass dummy spacecraft model as a fixture.

During this subsystem testing, a thruster vibration problem of mercury propellant leaking through the porous tungsten vaporizer plug was discovered. It was thought that vibration-induced pressure surges (similar to "water hammer") were the cause and some new designs were tried. These designs included using one or two snubbers in the mercury line to add damping. These snubbers were replaced by an annular orifice to increase damping. This orifice was later combined with two porous tungsten vaporizers in series.

These designs were not completely satisfactory and the vibration problem had not been solved when a problem arose of leakage occurring after short operation of thrusters that had not been vibrated. It was then determined that microscopic cracks had developed in the porous tungsten vaporizer plug when the plug was electron beam welded into the tantalum feed tube. This cracking tendency was alleviated by using a higher density porous material like that successfully used in the neutralizer system. A larger diameter plug was then required to obtain the proper flow rate. It was also determined that a small ullage space was required in the mercury reservoir so that expansion of the confined mercury, when heated, would not force the liquid mercury through the porous plug.

This combination of the higher density porous material and the allowance of the small ullage space in the mercury reservoir solved both the vibration and the operation problems.

Thermal vacuum testing. - Prototype thermal vacuum qualification testing was performed in two tests. The first was done without an operating thruster and the second had an operating ion thruster installed. Power conditioning for the thruster on the second test was supplied by an experimental unit mounted on the spacecraft. Prototype power conditioning qualification testing was performed as a separate test on a component basis as well as on the prototype spacecraft.

Thermal vacuum testing without thruster. - The purpose of the first part of this test was to soak and operate the spacecraft systems at the extremes of the temperature distribution ranges expected in flight. The test started by conditioning the spacecraft to the hottest temperature distribution predicted for flight. The electrical systems were turned on and operated in this mode for 48 hours. Systems were turned off and the temperature level was reduced to the lowest expected conditions. Systems turn-on and operation occurred at this low temperature and the cold soak was maintained for 48 hours. During this time, all spacecraft squibs were blown. This cold soak and operation was followed by a 72-hour period of hot soak and operation.

The test objectives were generally met; however, some failures were discovered in the test. The failures are summarized in table V.

TABLE V. - FAILURES IN PROTOTYPE SPACECRAFT -

THERMAL VACUUM TEST WITHOUT THRUSTERS

Failure	Correction
Emergency power system fuse blown	None - not attributed to vacuum test
Sweeping beam probe failed to stop	Bracket redesigned
Beam probe data output decayed slowly on turn-off	Capacitor removed from circuit
Thermal control coatings flaked off	Panel material changed from magnesium to aluminum

The spacecraft emergency power system failed to operate because of a blown fuse. This is a backup system which permits part of the thruster solar array to be switched to support the housekeeping array in event of emergency. The blown fuse was traced to a previous power conditioner failure that had resulted in a high current short. It was surmised that this failure also blew the emergency power system fuse, but it was not detected until this test; therefore, it was not a failure that could be attributed to the thermal vacuum test.

Two failures occurred in a sweeping beam probe assembly. The probe failed to stop at the end of travel limit switch. This was resolved by redesigning the bracket. Also, the telemetry output of the probe decayed slowly when the unit was turned off. The corrective action was to remove a capacitor from the signal conditioning circuit. Neither of these failures were a direct result of the thermal vacuum environment.

The last failure was evidenced as thermal control coatings which had flaked off the skins during the test. The panels were replaced and testing continued; however, the failure led ultimately to the use of aluminum rather than magnesium panels and better process control in applying the thermal coating.

Thermal vacuum test with operating thrusters. - The second portion of the thermal vacuum qualification test was a flight simulation test. One phase simulated the conditions of the first 20 hours in orbit; a second phase included startup of thrusters 1 and 2, in turn. The third phase was a reacquisition simulation. This exercised a backup spacecraft system which operates in the event the spacecraft tumbles and loses solar array power. All nonessential loads are automatically turned off and the essential loads are supported on a spacecraft battery which is not normally active in the system. The last test phase was a restart of thrusters 1 and 2, in turn.

During this test, several failures, as itemized in table VI, did occur. In the first part of the test, the space probe hot-wire filament failed to light. This was determined

TABLE VI. - FAILURES IN PROTOTYPE SPACECRAFT -

THERMAL VACUUM TEST WITH THRUSTERS

Failure	Correction
Space probe filament did not light	AGE guard designed to protect filament during handling of spacecraft
Cathode failure, thruster 1	Cathode redesigned
Two subcommutators damaged by high-voltage transients	Filter box designed for data system
Thermal coatings flaked off reflector erosion experiment	Coatings and processing changed
Hermetic seal failed on tape recorder	Seal replaced by wider soldered seal
Command system would not perform at low input voltage	Diode replaced

to be due to a filament which had been broken upon spacecraft installation in the tank. An AGE guard was designed to protect the filament during handling of the spacecraft.

Another failure was a cathode failure of thruster 1; however, the thruster did operate successfully after extended preheat. Thruster 2 operated properly and was shut down. Upon restart of thruster 1, two subcommutators were damaged due to relatively high voltage transients on the data input lines to the devices. The thruster problem was corrected by redesigning the cathode. The telemetry damage problem precipitated the inclusion of a filter box in the system to filter these transient spikes. This device was further evaluated on the experimental spacecraft.

A failure of thermal coatings flaking off the reflector erosion experiment (REX) was found in this test. It was corrected by changing coatings and processing techniques.

The qualification testing did expose these design deficiencies, which were corrected. All other components passed the tests satisfactorily.

After completion of the prototype qualification testing, all components were removed from the spacecraft for detailed visual inspection and bench checkout. This uncovered two more problem areas. A seal on the hermetically sealed tape recorder had leaked in vacuum, and upon reexposure to air pressure the recorder housing deformed enough to rub on the recorder mechanism and to prevent the recorder from operating. This recorder would have continued to operate in the vacuum but not in air; however, its vacuum life would have been limited due to lubricant deterioration at the very low pressures. This problem led to incorporation of a wider band soldered seal installed with better quality control.

Another problem was the failure of the command system to operate down to its specified low input voltage. This was corrected by changing a diode in the circuit. This was a design deficiency not caused by the thermal vacuum test.

This completed the qualification test program on the prototype spacecraft, with most components successfully being qualified. The problem areas described were resolved and qualification was completed on a subsystem basis. Because it is a key component, the power conditioner vacuum qualification is described in the next section.

Power conditioner qualification test. - The qualification model power conditioner was mounted on the experimental mass dummy spacecraft for vibration and shock, and then mounted on the prototype spacecraft for thermal vacuum qualification. In the thermal vacuum test, it was baked out at 165° F (74° C) for 12 hours to simulate flight conditions. It was then subjected to five startups to include extremes of input voltage and temperature as follows: (1) high voltage, high temperature; (2) high voltage, low temperature; (3) nominal voltage, high temperature; (4) nominal voltage, low temperature; and (5) low voltage, low temperature. In this test, the high, nominal, and low voltages were 75, 66, and 54 volts dc and the high and low temperatures were $+110^\circ$ F (43° C) and -25° F (-32° C). All possible modes of operation were successfully demonstrated by the unit.

In addition to this qualification on the spacecraft, the prototype power conditioner was qualified on a component basis by operating at temperatures as high as $+140^\circ$ F (60° C) on the baseplate and as low as -35° F (-37° C). The input voltage was varied from +50 to +75 volts dc.

Thermal design evaluation. - In order to make a final evaluation of the thermal control design of the spacecraft, 142 thermocouples were installed on the prototype unit. A test was performed which allowed the spacecraft temperature distribution to attain equilibrium in a variety of power dissipation and heat input cases. It was shown that the temperature distributions predicted by the analytical model were on the average within $\pm 5^\circ$ F ($\pm 2.8^\circ$ C) of the test values in all cases. The test also demonstrated that temperatures indicated by thermistor circuits read through telemetry were very close to thermocouple readings obtained via hardline.

Endurance testing. - The prime objective of the SERT mission was to operate an ion thruster system in space for 6 months (4380 hr); therefore, it was necessary to demonstrate appreciable endurance capability of the spacecraft before launch by extensive ground testing in the thermal vacuum environment. The prime vehicle for this testing was the prototype spacecraft which permitted accruing time on all spacecraft components including the ion thruster system in a single test installation.

Supplementary life tests were performed on two power conditioner - thruster subsystems in contractors' vacuum tanks. These component tests were used to evaluate the complete ion thruster integrated system, but especially the areas of the main cathode and the neutralizer. These long duration tests provided information on the thruster

system characteristics which change with operating time. Compatibility of the integrated system in such areas as materials suitability and power conditioner - thruster control loop stability was shown, thus providing confidence in the design capability.

A supplementary life test was also carried out on a tape recorder. This device has many mechanical features subject to wear which required confidence testing. There are two tape recorders on the spacecraft, operating alternately, so one unit potentially requires only 3 months life.

At the time of launch, the prototype spacecraft had accrued over 3500 operating hours of which almost 2400 hours was with an operating thruster. This prototype spacecraft endurance test was continued as long into the program as possible. It was terminated by required shipment of the spacecraft to the launch base to support first integration tests with the launch vehicle. This coincided with the time that the flight spacecraft was ready for installation into the vacuum facility. The two power conditioner - thruster life tests had accumulated 1410 and 1508 hours, respectively, at launch time. Both these tests were still in process at launch day. A single tape recorder demonstrated 4000 hours life without failure prior to lift-off. This testing gave confidence that the design had potential to perform for the duration of the mission.

Flight Program

The flight spacecraft was assembled using components which had completed a flight acceptance test program on the component level. The basic philosophy was that this spacecraft model was to be tested for sufficient time and at sufficient levels to prove it ready for flight; however, it was not to be tested such that component lifetimes were in jeopardy. Of particular importance, the total operating time of the two ion thrusters had to be limited to about 100 hours maximum. This was required to minimize the amount of sputtered material accumulated between the thruster screen and accelerator plates. Upon exposure to air, heavy deposits there can peel off and cause short circuits. Also, the mercury propellant usage was kept low so that refilling before launch was not required. Thruster operation on the spacecraft also caused contamination of some nonreplaceable thermal control surfaces, which was another reason to limit the operating time. It was a basic philosophy to minimize spacecraft disassembly and handling during and after the qualification program.

Vibration and shock tests. - After initial electrical integration testing, the spacecraft, complete with squibs and pressurized gas sphere, was vibration tested in the same way as the prototype spacecraft and in the same facility. The difference was that the flight model was subjected to only two-thirds of the peak stress levels used on the prototype. The specific levels used are in tables I to IV. The test environment was at

least as severe as the vehicle was predicted to produce and no spacecraft failures were produced as a result of vibration and shock tests.

Thermal vacuum test. - The thermal vacuum portion of the flight acceptance test was conducted on the complete flight spacecraft with squibs in place. The squibs were fired during the course of the test and replaced later. The test started with a period of 12 hours of spacecraft and power conditioners outgassing, followed by spacecraft turn-on. During the course of the test, all systems were checked at high and low temperatures. Each power conditioner - thruster system was subjected to one high-temperature, high-voltage start and one low-temperature, low-voltage start. The test time was limited so that only 100 hours of thruster operation time was accumulated during the test. This limited the amount of contamination from deposited mercury and sputtered target material to a low level. While the outer thermal control surfaces on the spacecraft were to be replaced at the launch base, much of the exposed spacecraft structure was not replaceable so contamination had to be minimized.

The test was generally successful; however, two failures did occur. One of the power conditioner telemetry channels read erroneously. Investigation revealed it to be a random failure of an operational amplifier in the power conditioner which affected data only and not the functioning of the unit. The amplifier was simply replaced. Also, a thermistor measuring a structure temperature gave erroneous readings. It, too, was replaced and the problem was corrected.

Communications network compatibility test. - The in-orbit tracking, data acquisition, and command support for the mission are supplied by the Satellite Tracking and Data Acquisition Network (STADAN) controlled by NASA Goddard Space Flight Center (GSFC). A compatibility test was performed using the flight spacecraft and a NASA-GSFC communications van to evaluate the command threshold and telemetry signal strength margin. The tests resulted in the correction of two anomalies. The degree of carrier modulation was reduced to allow more reliable minitrack tracking using the telemetry signal carrier.

The tests also resulted in interchanging a flight transmitter with a backup unit to achieve better telemetry coverage. The original flight transmitter had phase instability which, while it would have been usable in flight, would have resulted in reduced signal margin. A flight backup transmitter thus became the prime flight unit.

Preshipment testing. - The completion of thermal vacuum testing marked the end of the opportunity to operate the flight thrusters and power conditioners as a system before launch. The remaining testing then was limited basically to communications and power system checks and checks of auxiliary experiments. The post-thermal-vacuum testing consisted of replacing squibs fired in the vacuum test, and making final alinement checks on the ion thrusters. An electrical checkout of all systems except the thrusters and power conditioners was carried out and the spacecraft was put into the

shipping container for truck transport from Cleveland, Ohio, to the launch base at Vandenberg Air Force Base, Lompoc, California.

Launch Base Testing

Launch base testing with prototype spacecraft. - Launch base testing was begun using the prototype spacecraft for the first integrated tests on the launch pad with the launch vehicle. These tests demonstrated that the spacecraft support AGE was satisfactory to support preflight testing. They also verified proper functioning of the spacecraft-vehicle electrical interface. It was desirable to use the prototype unit at this point in order to allow a later range arrival date for the flight spacecraft, and also to lessen the exposure of the flight spacecraft to the pad environment.

Launch base testing in spacecraft laboratory. - When the flight spacecraft arrived at the range, it was checked out in the spacecraft laboratory area. The basic philosophy was to check out only those systems which did not require disassembly of components or of connectors which had been validated in the vibration and vacuum tests and subsequent checkouts, unless the original integrity could be reestablished. Disassembly was allowed in the case of the power conditioners which power the thrusters. It was allowed here because of the criticality of the system and the time (10 days) that had elapsed since last system operation in the vacuum chamber. The spacecraft had also been packaged and truck transported across the country and unpacked in the interim period. Relatively good accessibility of the harness connections made reestablishing the system integrity possible. This accessibility was provided by the use of "open type" connections. Each high-voltage lead was attached to the power conditioner via an individual ceramic-insulated stud feedthrough. This permitted thorough inspection after reassembly and a confident reassessment of reliability. The 13 lug and terminal connections of the wire harnesses which connect each power conditioner to its thruster were removed at the power conditioner end and a dummy load was connected. This dummy load simulated the thruster by providing resistive loads for the power conditioner power supplies so that the power conditioner could be tested under load with an open-loop check of the control points. After a dummy load check, the load was disconnected and the thruster harness reconnected to the power conditioner. This was done on both ion thruster - power conditioner systems. This last connection could not be validated by system operation before launch; however, it was successfully done by careful wire labeling and inspection of the installation.

In addition to the dummy load tests in the spacecraft laboratory, electrical tests of the communications and power systems were carried out. The squib bridgewire resistances were rechecked and connected to their harness connectors. The titanium high-pressure-gas sphere of the Backup Acquisition Control System was pressurized to

3600 psi (2.48×10^7 N/m^2) with Freon-14 and pneumatic system leak checks were performed. The flight spacecraft was replaced in the shipping container for movement to the pad and assembly with the launch vehicle.

Launch base testing on the pad. - Testing after Agena vehicle mate was limited to simple communications checks to show the spacecraft was ready to support integrated testing with the vehicle. Integrated testing was limited to verifying that the spacecraft-vehicle electrical interface was functioning properly. This interface functionally consisted of power cables routed from the solar array on the Agena vehicle aft rack into the spacecraft. Also, a ground cable in this bundle connected the Agena ground point to the spacecraft ground point. A connector at the interface carried the power required to operate the Agena vehicle horizon-sensor system from the spacecraft power system as well as carrying the horizon-sensor telemetry signals from the vehicle to the spacecraft. Another connector at the interface carried the instrumentation lines from the six solar array thermistors and three control cells to the spacecraft. The third interface connector carried three commands from the spacecraft command system into the Agena vehicle. Also, it carried lines from the spacecraft to the Agena vehicle umbilical connector to allow spacecraft pad operation and control using ground power and equipment. After integrated testing demonstrated this interface to be correct, a calibration of the vehicle horizon-sensor system, as read through spacecraft telemetry, was carried out. This completed the spacecraft preflight checkout needs.

During the countdown, a simple test was made to show that the command and telemetry systems were still operable. The control moment gyros were spun up on ground power so that the rotors were spinning during ascent. That completed ground checkout and preparation for launch. At the time of launch, the flight spacecraft had accumulated 775 hours of testing, of which 53.5 hours was with thruster 1 and 53.5 hours was with thruster 2 for a combined total of 107 hours with thrusters operating.

Initial Flight Results

As a result of this extensive test-and-development program encompassing vibration, thermal vacuum, and endurance testing, the SERT II spacecraft was successfully launched on February 3, 1970. It successfully resisted the launch environment and has performed satisfactorily in the thermal vacuum environment of space. Six-month endurance capability of the prime experiment, the ion thruster systems, remains to be demonstrated.

The first ion thruster was successfully started on February 11, 1970. Startup consisted of operating the thruster at the 30 percent thrust level and monitoring the vehicle attitude. Thrust misalinement can produce large offsets in this gravity-gradient-stabilized satellite. No large offsets occurred, so the thrust level was raised to 80 per-

cent and thruster operation and spacecraft attitude again monitored before operating the thruster at full thrust. Successful thruster startup with no required gimbal corrections was achieved. The thruster was shut down on February 12, 1970, according to the flight plan. The second thruster started successfully with no required gimbal corrections on February 14, 1970. This thruster was shut down prior to the solar eclipse on March 7, 1970, which interrupted solar array power; however, it was restarted on that same day to continue toward the goal of 6 months of operation. Several reports have been presented which further describe the spacecraft and the preliminary flight results. These reports are listed as references 5 to 12.

Program Timing

The total SERT II program from the time of approval through launch required 42 months. Figure 11 contains the milestone schedule. It shows that the flight spacecraft was assembled 15 months before launch; however, the power conditioners were not available for integration into the system until 2 months before launch. During that time, development problems of severe arcing inside of the power conditioner were being solved. Reference 8 discusses those efforts in detail. During this time, the prototype spacecraft was in endurance test.

CONCLUDING REMARKS

Development and testing of this electrically propelled spacecraft was carried out in a manner typical of most spacecraft projects. Component qualification and system qualification to expected shock, vibration, thermal, and vacuum environments were carried out. In addition, some special provisions were required to cope with the conditions resulting from the mercury bombardment ion thruster and its required power conditioner. All testing with operating ion engines was performed in a vacuum chamber. A contamination shield was required to protect the spacecraft from sputtered target material in the relatively small size of the chamber.

The high voltages developed in the power conditioner necessitated vacuum soak times of more than 12 hours after exposure to air. This was to allow complete outgassing before turn-on to prevent arc-over. Also, the high voltages involved and the need to electrically isolate the spacecraft from ground governed the ground power supply system design which included isolating the supplies from ground and connecting them to the commercial power lines through isolation transformers. To simplify the isolation problems, control of the spacecraft and the reception of data from it were accomplished by a radiofrequency command and data link to reduce the number of hardlines to the vehicle.

The mission requirement of a 6-month endurance test of an ion thruster system dictated that its long life characteristics be first established by extensive ground thermal vacuum tests. To accomplish this, three thruster life tests were initiated - two with integrated power conditioners on a subsystem basis and one utilizing the complete prototype spacecraft system as a test installation. Results from these tests provided the systems confidence required for flight go-ahead.

This program was accomplished in 42 months from time of approval until launch by assembling three operational spacecraft and a mass dummy model. The mass dummy was used to evolve a vibration- and shock-resistant system. The operational experimental model was used to demonstrate feasibility and systems compatibility and to develop assembly and test techniques for the prototype and flight models. The prototype model was identical to the flight model but it was subjected to more severe testing to demonstrate design adequacy. The flight model testing was limited in severity and duration so that a minimum of useful life was expended in ground tests. There was no backup spacecraft built but rather backup units for all replaceable components were prepared and were ready for installation, if needed.

TABLE VII. - OPERATING TIME ACCUMULATED ON VARIOUS

EQUIPMENT AT TIME OF LAUNCH

	Total hours	Hours in vacuum	Hours with thruster
Experimental spacecraft	500	415	375
Prototype spacecraft	3500	3200	2400
Flight spacecraft	775	544	107
Ion thruster - power conditioner:			
Life test 1[a]	1410		
Life test 2[a]	1508		
Tape recorder (endurance test)	4000		

[a]Tests still in progress at time of launch.

Table VII summarizes the significant operating hours accrued on various systems at the time of launch. Ion thruster - power conditioner life tests 1 and 2 were still in process at the time of launch. This program culminated with a successful launch and start of spacecraft operation on February 3, 1970.

Lewis Research Center,

National Aeronautics and Space Administration,

Cleveland, Ohio, November 4, 1970,

704-13.

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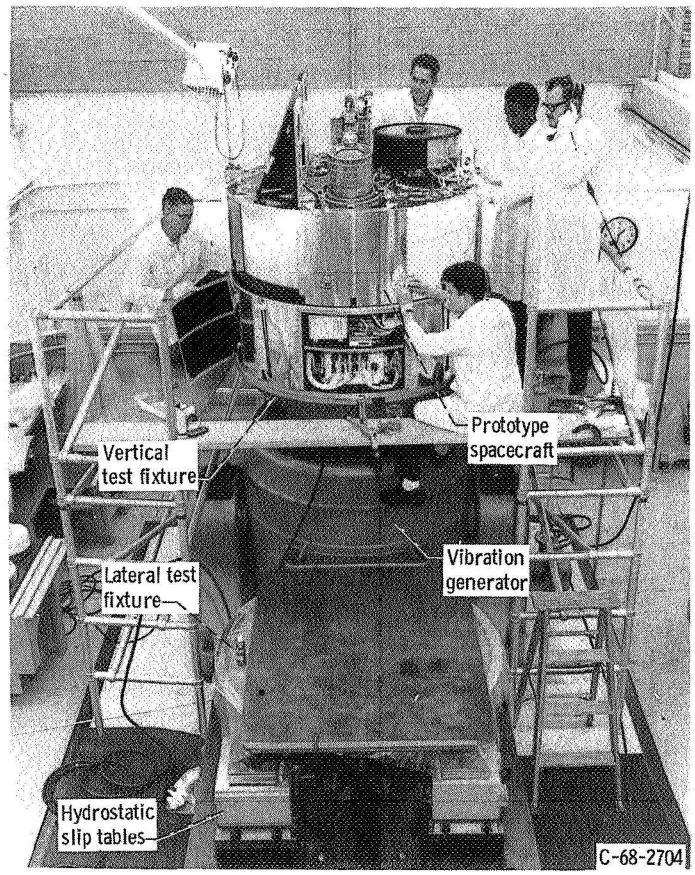

Figure 1. - Prototype spacecraft mounted on vibration generator in preparation for thrust-axis vibration.

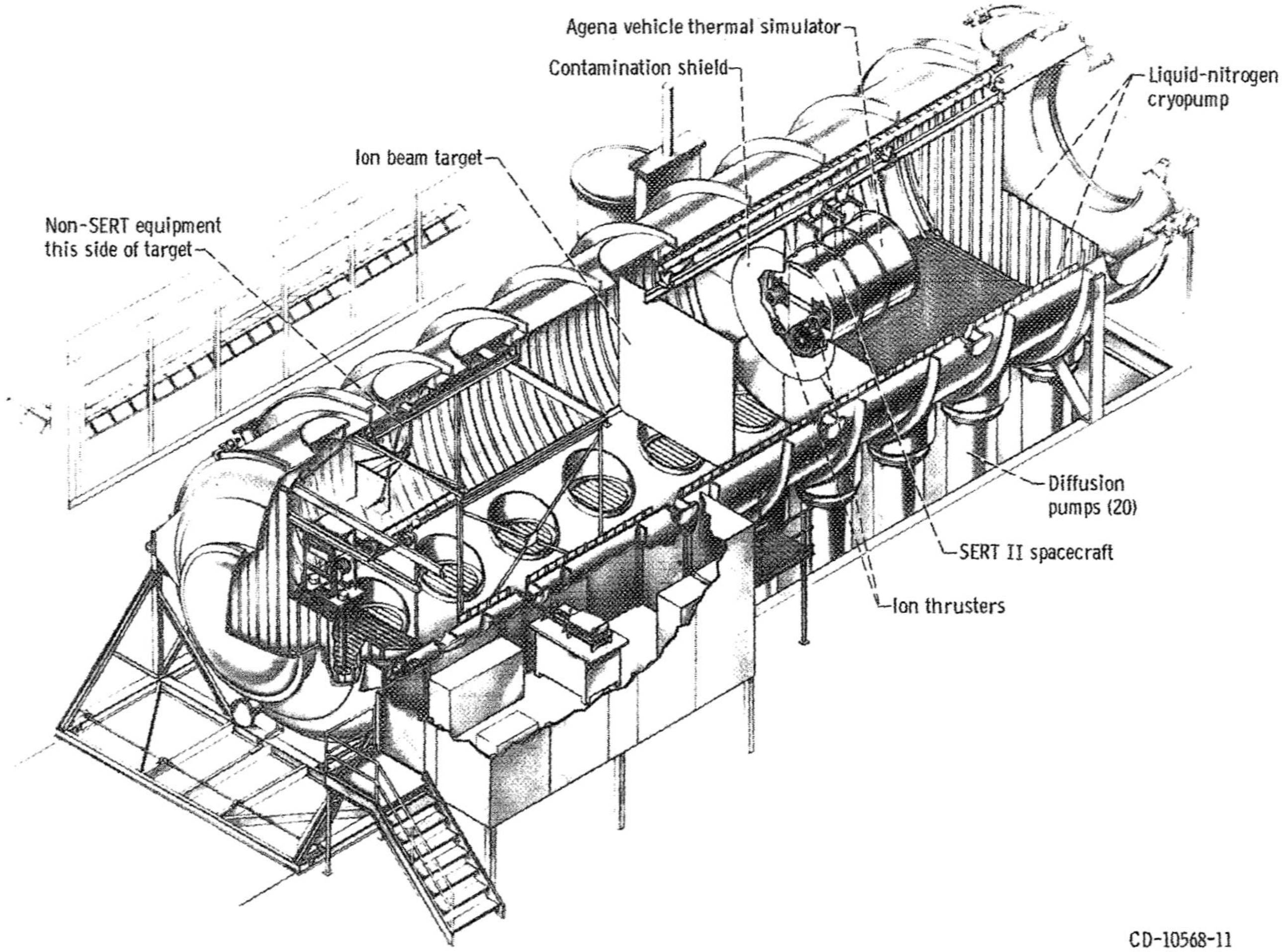

Figure 2. - SERT II spacecraft mounted in vacuum facility.

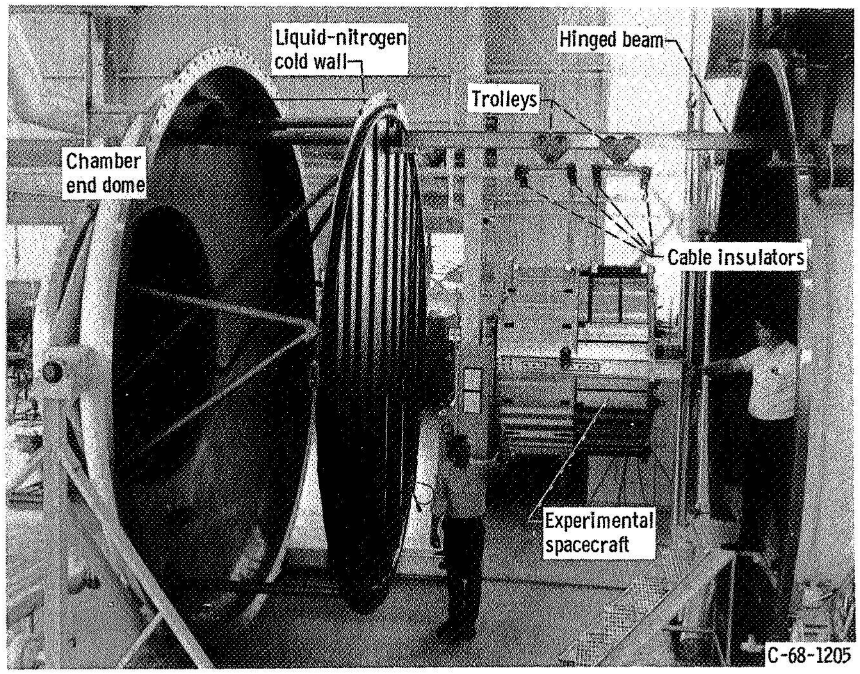

Figure 3. - Experimental spacecraft being installed in vacuum facility.

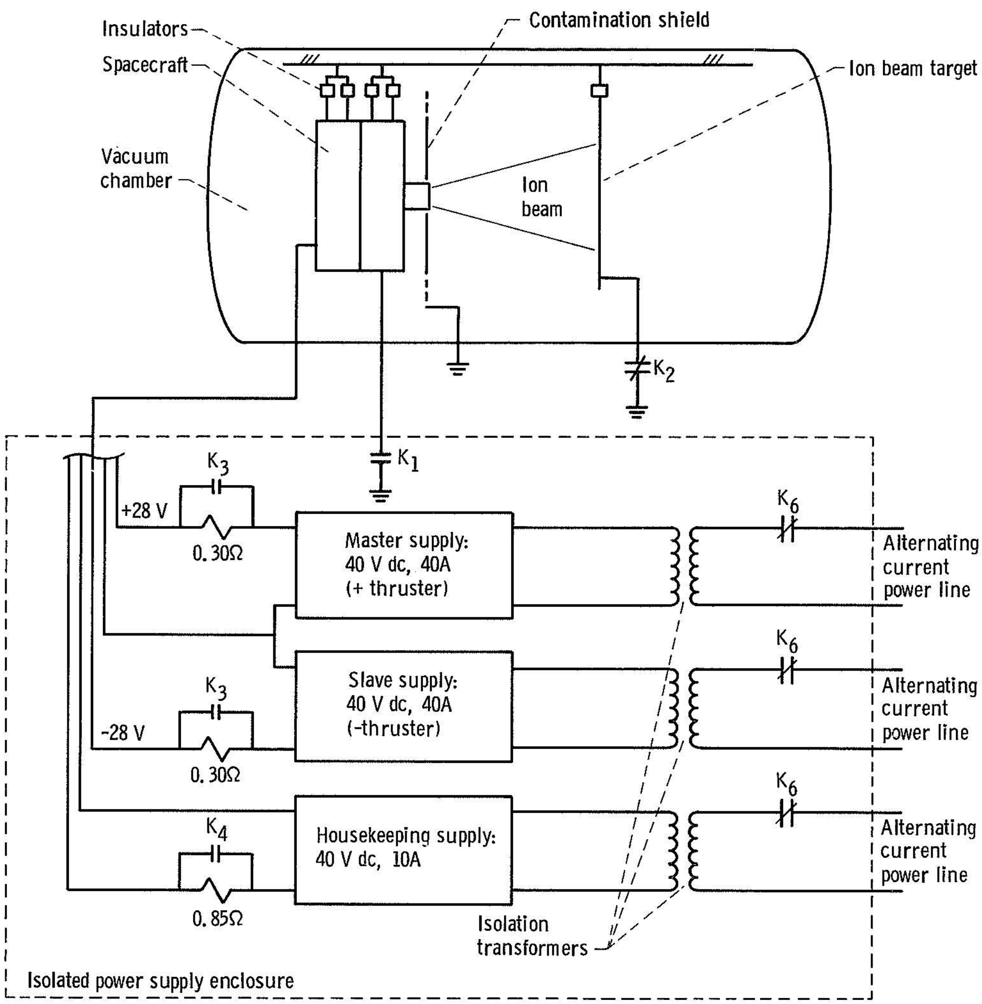

Figure 4. - Spacecraft vacuum test electrical installation.

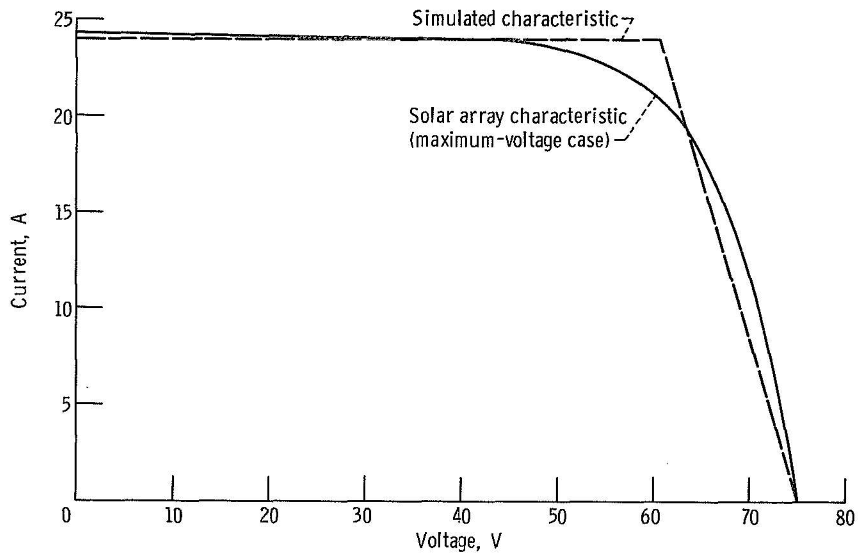

Figure 5. - Thruster solar array and simulated characteristic.

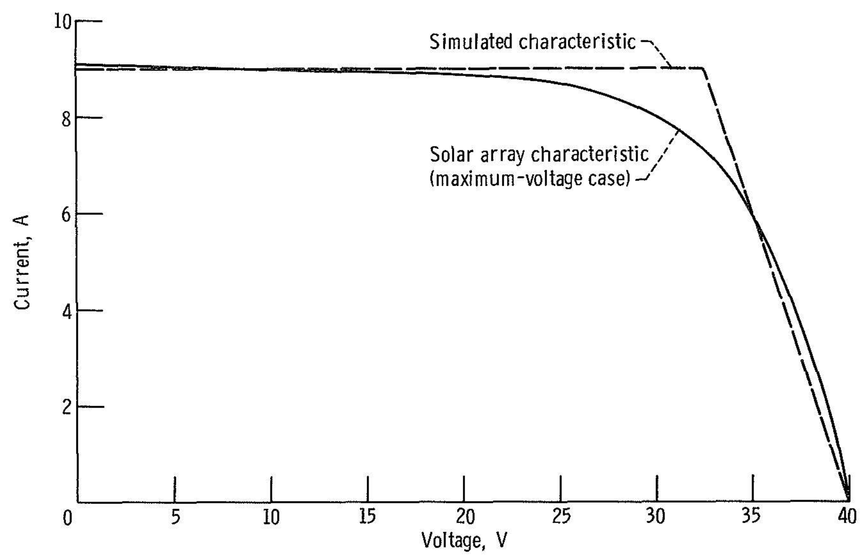

Figure 6. - Housekeeping solar array and simulated characteristic.

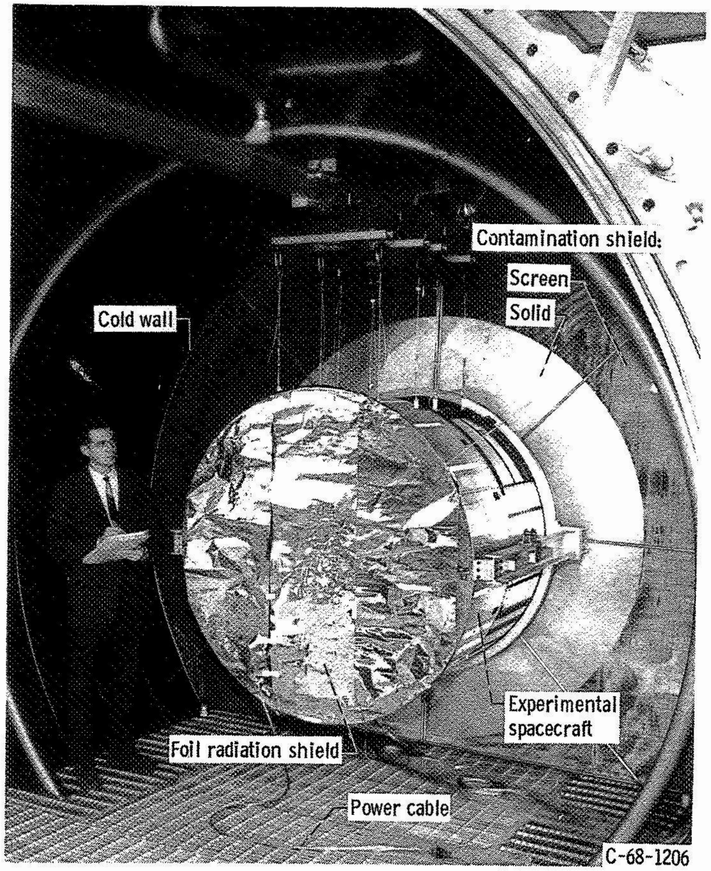

Figure 7. - Experimental spacecraft mounted in vacuum chamber - without Agena vehicle thermal simulator.

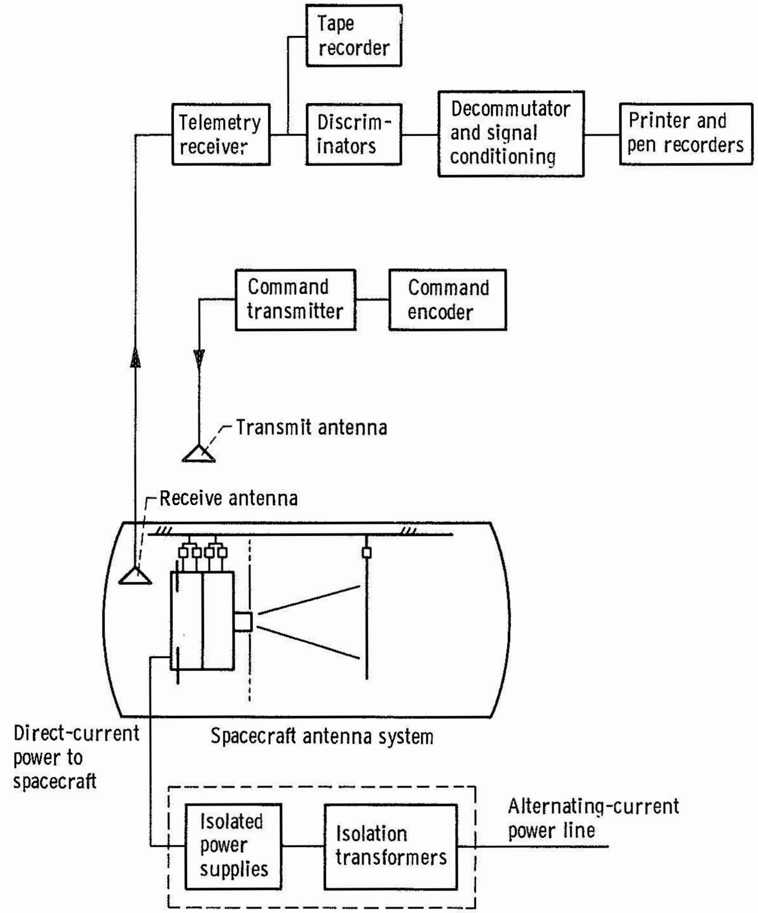

Figure 8. - Spacecraft test installation for radiofrequency link testing.

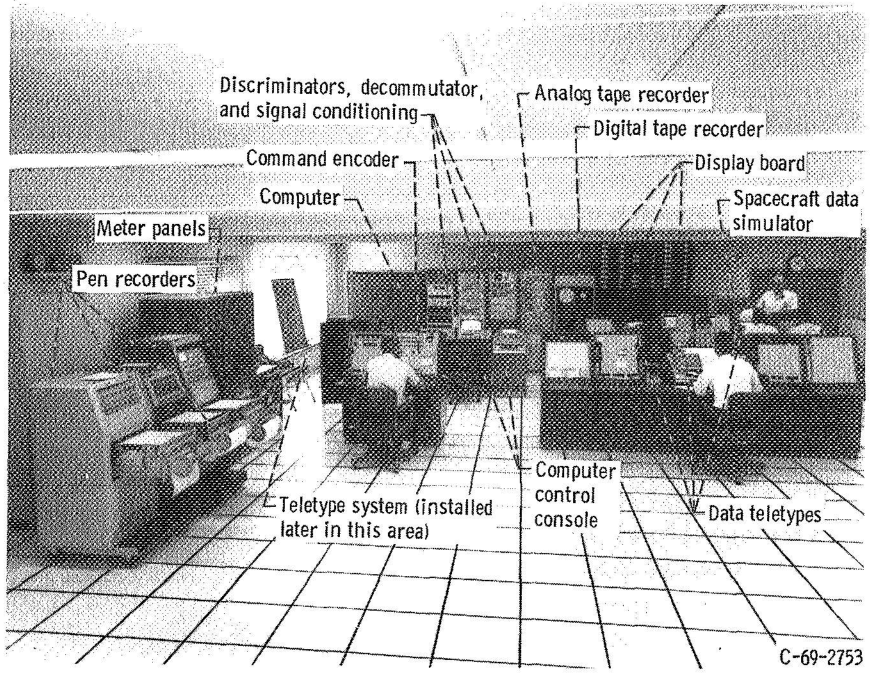

Figure 9. - SERT II control center.

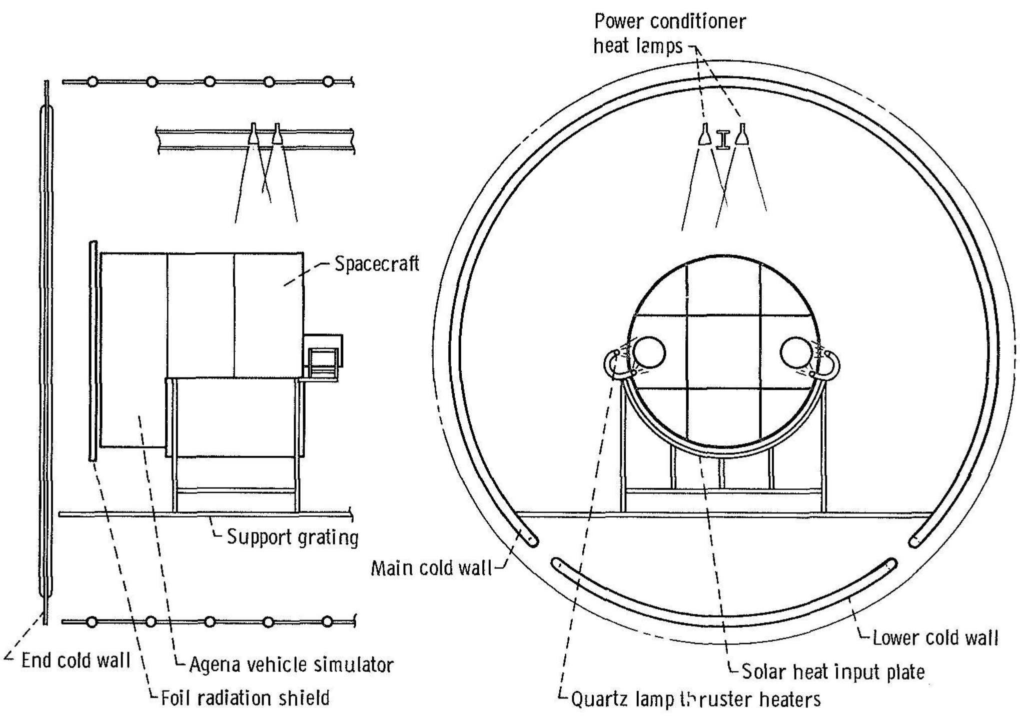

Figure 10. - Spacecraft vacuum test thermal-control-system configuration.

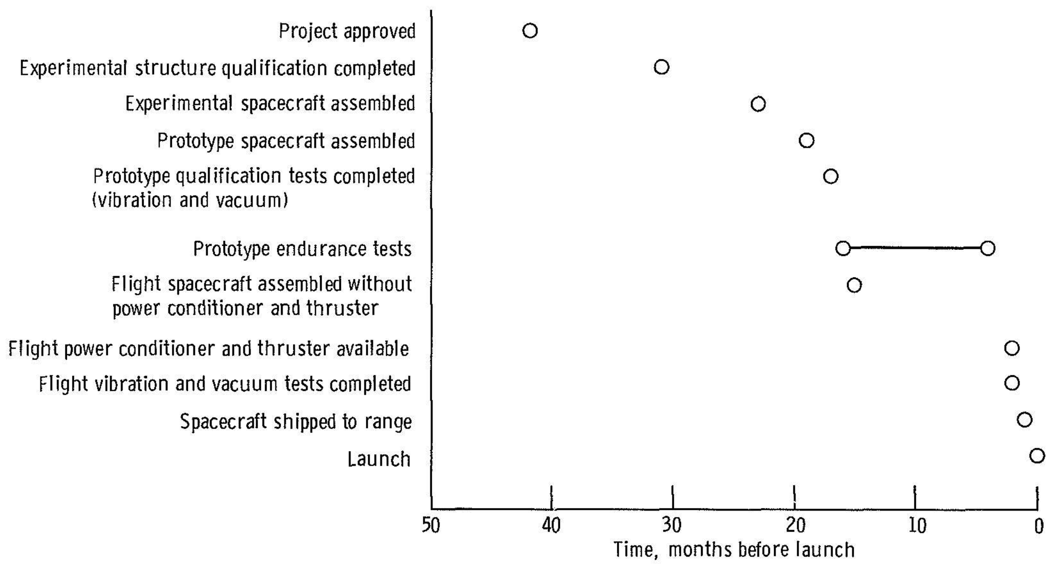

Figure 11. - SERT II project milestones.

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